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C68-5307

MARINER R SPACECRAFT FOR MISSION P-37/P-38 [U]

EPD - 46, Rev. 1

9/28/61

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JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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SECTION I

INTRODUCTION

A. PURPOSE

Considering the limited time available to the Mariner R project this document has been prepared to acquaint individuals associated with the project with the spacecraft and its mission.

The document is to provide a general description of the existing Mariner R spacecraft design as it exists in September 1961. This document is in no way to serve as a specification, but is generated merely to provide non-contractual, general information. Definitive and contractual requirements are furnished in such official documents as the Vehicle Integration Specification, Spacecraft Design Specifications, etc.

The Mariner R (Agena) project is aimed at providing a P-37 and P-38 mission capability to Venus in 1962 as a replacement for the Mariner A project which was cancelled during September 1961 due to unavailability of Centaur launch vehicles.

B. MISSION OBJECTIVES AND DESIGN GOALS

The primary objective of the Mariner R project is to develop and launch two spacecraft to the near vicinity of the planet Venus in 1962, to receive communications from the spacecraft while in the vicinity of Venus, and to perform a radiometric temperature measurement of the planet. A secondary objective is to make interplanetary field and/or particle measurements on the way to Venus and in the vicinity of Venus.

The launch vehicle to be used in this project is the Atlas Agena B, providing a spacecraft weight of approximately 460 pounds for at least a 47-day launch window. It is planned to launch two probes sequentially off the same pad.

This spacecraft is intended to evolve from and to benefit from the experience gained in the Ranger and Mariner A projects. Many of the components to be used in these flights are modifications of existing Ranger equipment. In addition, equipment originally planned for Mariner A is to be incorporated in those areas where the Ranger components cannot meet those needs of the mission. The time scale is such that presently designed equipment should be used even to the extent of reducing the mission objectives.

In planning the mission, certain goals were established initially as follows:

1. Accomplish the mission in the limited time available.
2. Use the present Agena B unmodified.
3. Base the spacecraft design upon Ranger 3 concepts.
4. Achieve a design weight that is compatible with both the mission demands and the firing window.
5. Use redundancy wherever possible to minimize sequential equipment and operation necessary to accomplish the mission.
6. No sterilization required by designing for 1/1000 probability of planetary impact.

In some cases these goals can be met, in other cases the goals were not completely met. For example, there is a high confidence that two spacecraft can be prepared in time to meet the 1962 firing window for Venus. The design has drawn heavily from presently developed hardware for the Ranger and Mariner programs, thus it is believed that it would be possible to complete the detailed design, assembly, and test both spacecraft in time for the firing.

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The Agena B weight must be reduced and the parking orbit probably changed from 100 to 85 nautical miles to meet the mission requirements. Both of these appear feasible.

Spacecraft design based on Ranger 3 concepts can be accomplished if one recalls that Ranger 3 was designed with a bus and passenger concept whereby the basic mechanisms to achieve the mission were designed first, and then the instruments that were to be delivered at the destination were restrained to the bus capability. It was not possible, however, to use Ranger 3 hardware to the extent initially desired when the present mission was examined more closely. A detailed study of the ability of RA 3 hardware was made. Factors such as availability and reliability forced a modification of this approach. Table I-1 shows the sources of the hardware for the proposed design. Notice that the major elements of communications and guidance and control are available from the Mariner A program. The design for the Mariner A considered the long life required for a planetary mission while Ranger 3 requirements were satisfied within a few weeks.

It can be said that the functional philosophy and design are a continuation of the Ranger. However, in actuality this spacecraft is a reduced Mariner A in a new envelope.

A maximum allowable weight of 460 pounds was selected after estimating the spacecraft weight and examining the Agena performance capability. This provides approximately 40 pounds for scientific instrumentation. This weight provides 47 days of firing window based on certain feasible Agena modifications. To provide greater assurance of two launchings every effort should be made to minimize the time between launchings.

Redundancy of design although attempted was not achieved extensively. Weight considerations prevented redundant communications.

There is a period during the cruise mode where the sun-probe-earth angle approaches 180 degrees. The omni-antenna has become ineffective after approximately ten days. Thus, if the high-gain antenna is not pointing to the Earth because the earth seeker is locked onto a false target, no data will be received during this period.

Scientific instruments are operated during the cruise mode and their data telemetered to Earth except for the condition mentioned above.

In the vicinity of the planet the advantages of a space stabilized platform are exploited sufficiently to demonstrate that Venus had indeed been approached

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and measured.

It is important that planetary measurements be made prior to and during encounter. After encounter the reliability drops sharply. This is due to the possibility that the long range earth seeker may lock onto Venus since it is brighter and larger than the Earth.

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Table I-1. Hardware Sources for Proposed Design

	RA 1	RA 3	MA	New	Reworked
A/C		Sun Sensors Ant. Drive Gas System	L. Range Earth Gyro & Electronic Sw. Amp. Relay & Pwr. Sw. Ant. Servo Auto Pilot		
CC&S					RA 3 reworked
Power			Battery Converters	Solar Panels	
Comm.	Omni		Transponder Power Amp.	RF Adaption T/M	High gain ant. (MA) Command (MA) Pwr. Dist. Data Encoder (MA)
Systems				Cabling	
Eng. Mech.		Basic Hex		Ant. Yoke Sun sensor mount Omni support Science support Packaging	
Propulsion		Midcourse Prop. System			

SECTION II

DESCRIPTION OF MARINER R SYSTEM

A. SPACECRAFT DESCRIPTION

Model R Mariner Spacecraft will contain the following elements:

A system for planetary and interplanetary scientific measurements.

Solar power panels and power storage and conversion equipment.

Two-way communications equipment.

Environmental control equipment.

Modified Ranger attitude control based on tracking of the sun and earth.

Midcourse maneuver.

Instrumentation and data handling.

Modified Ranger structure.

Physically, the spacecraft resembles Ranger I except that the superstructure supporting the omni-antenna will be modified. Functionally it performs much like the Ranger I where standard events are controlled by a central clock and special events are controlled by real time commands. It bridges the gap between lunar missions and planetary missions by utilizing Mariner A developed hardware that is designed with long life in mind.

The gross configuration of the spacecraft is shown in Fig. II-1.

A block diagram of the spacecraft is shown in Fig. II-2.

Separation from the Agena is performed as on the Ranger. Solar acquisition is immediately achieved. The Long Range Earth Seeker originally planned for the Mariner A is used for earth acquisition and is aligned to the directional high gain antenna. Pointing of the antenna consists of proper spacecraft roll and antenna hinge orientation while remaining locked onto the sun. Until earth acquisition communication with the spacecraft must rely on the omni-antenna.

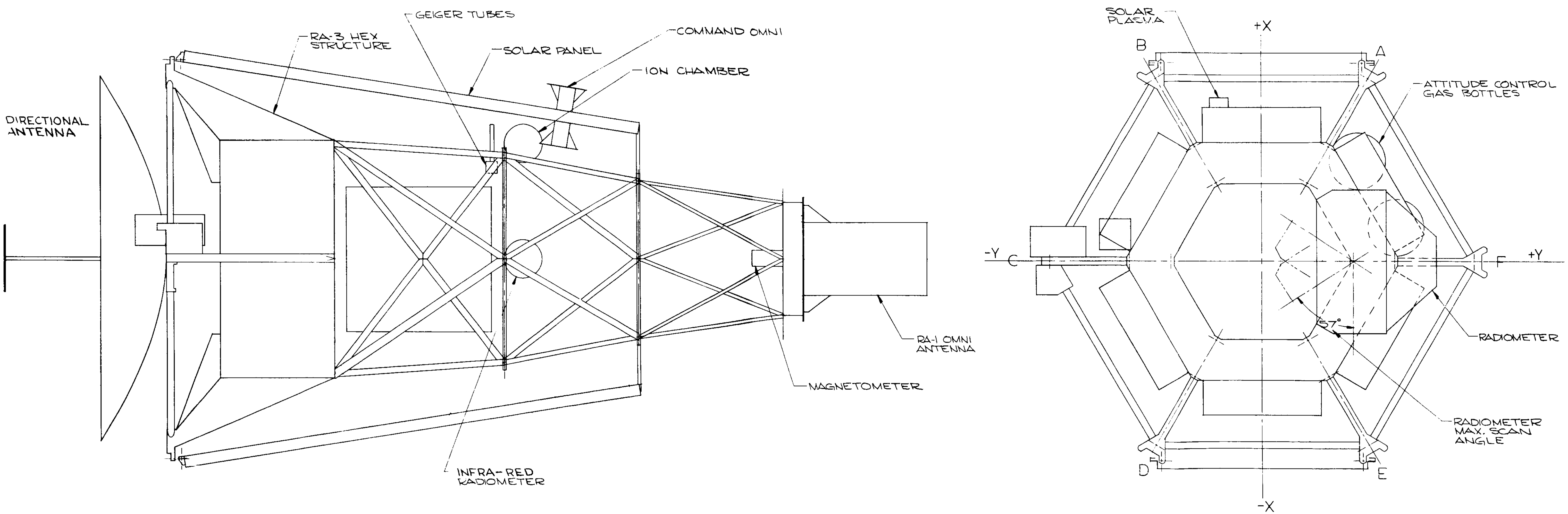
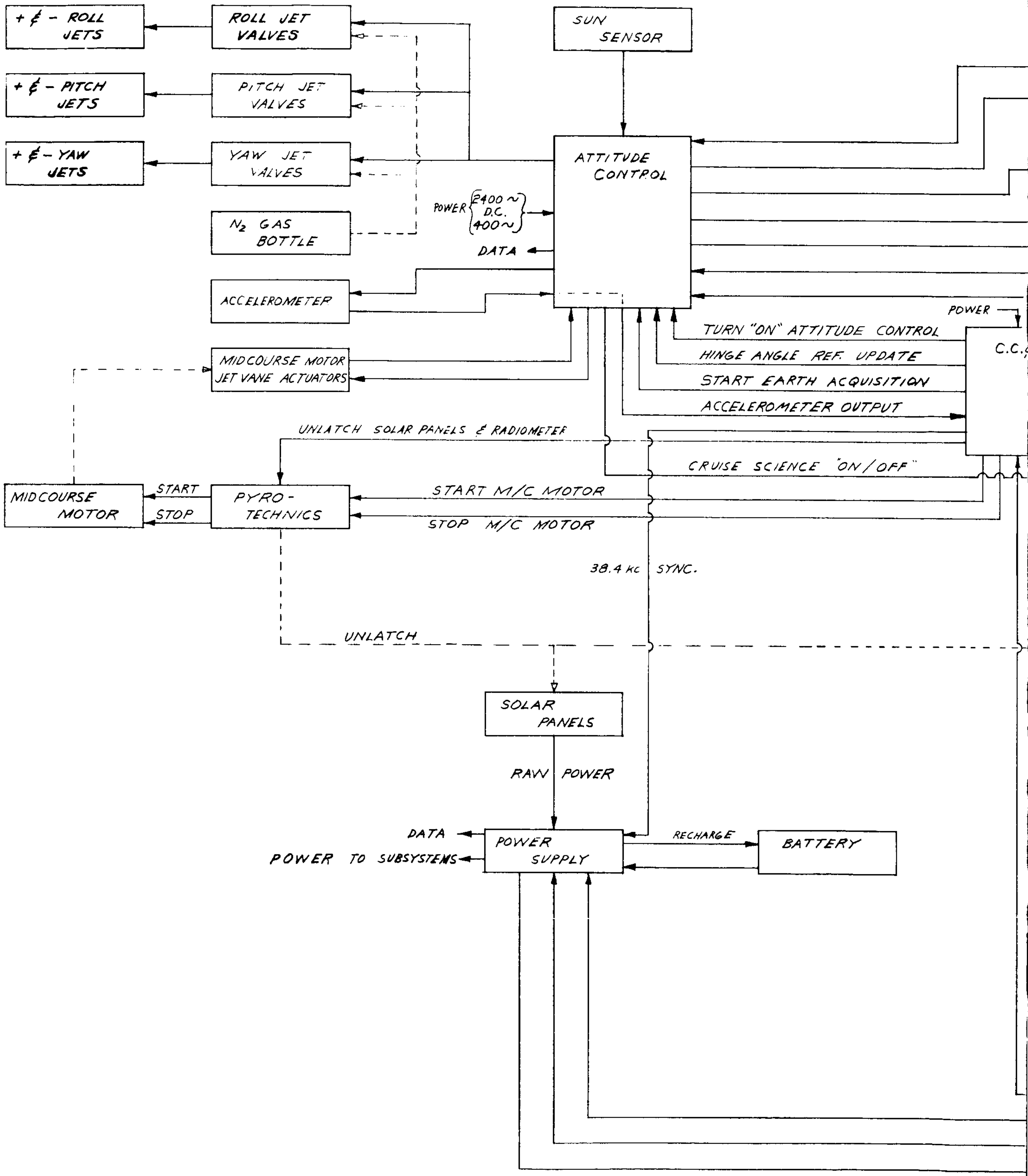


Fig. II-1. Mariner R Configuration



+ & - ROLL JETS
ROLL JET VALVES
+ & - PITCH JETS
PITCH JET VALVES
+ & - YAW JETS
YAW JET VALVES
N2 GAS BOTTLE
SUN SENSOR
ATTITUDE CONTROL
POWER 2400 ~ D.C. 400 ~
DATA
ACCELEROMETER
MIDCOURSE MOTOR JET VANE ACTUATORS
POWER
C.C.
TURN "ON" ATTITUDE CONTROL
HINGE ANGLE REF. UPDATE
START EARTH ACQUISITION
ACCELEROMETER OUTPUT
UNLATCH SOLAR PANELS & RADIOMETER
CRUISE SCIENCE "ON/OFF"
MIDCOURSE MOTOR
START
STOP
PYRO-TECHNICS
START M/C MOTOR
STOP M/C MOTOR
38.4 KC SYNC.
UNLATCH
SOLAR PANELS
RAW POWER
DATA
POWER TO SUBSYSTEMS
POWER SUPPLY
RECHARGE
BATTERY

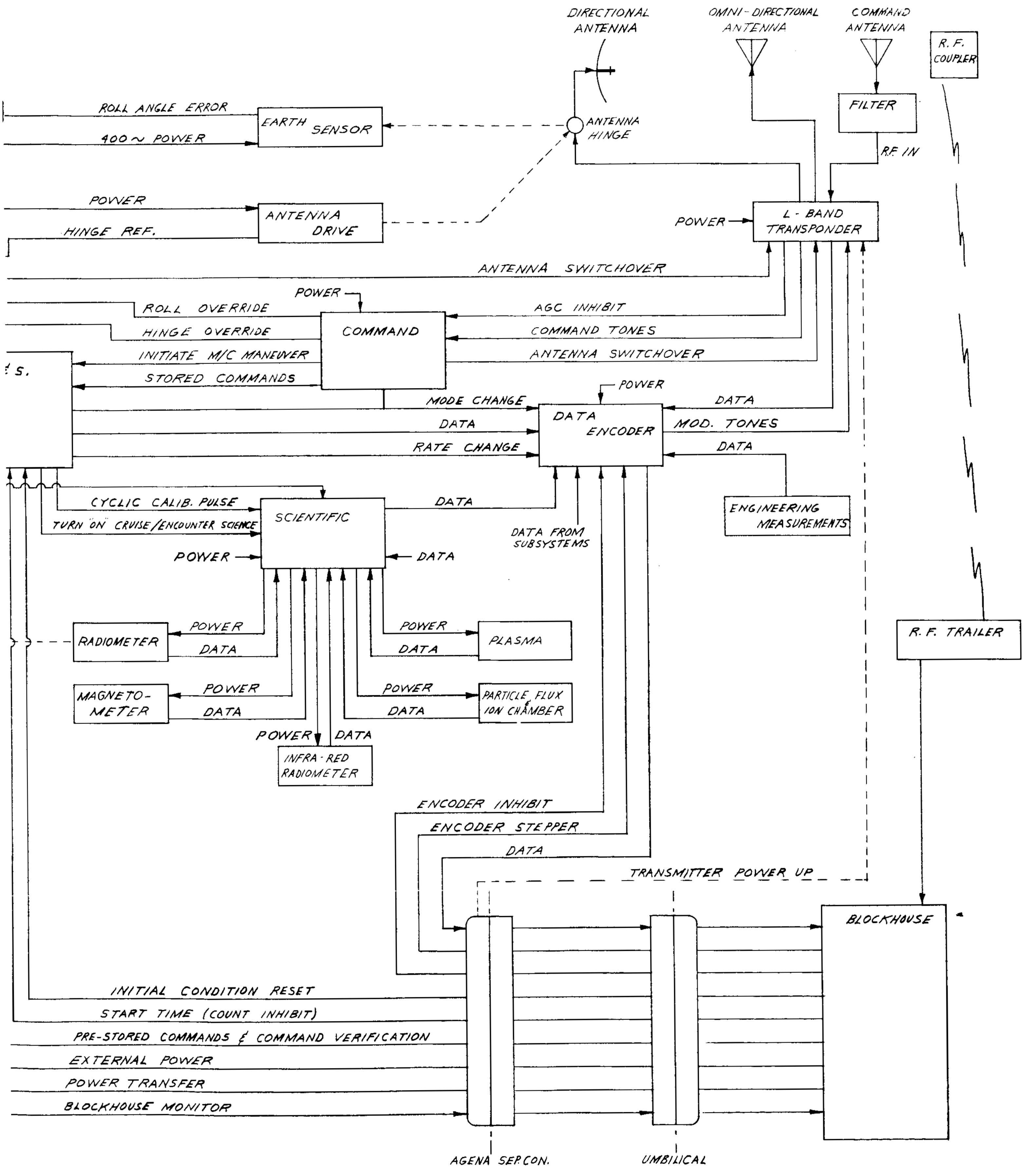


Fig. II-2. Mariner R, Block Diagram

B. FUNCTIONAL DESCRIPTION

1. Antennas

The antenna system will consist of an adaption of the Ranger 1 omnidirectional antenna, a modified Mariner directional antenna, and a Mariner A command antenna system. The Ranger 1 omniantenna is to be used because it has a greater spherical coverage than the RA-3 omniantenna. The Mariner parabolic directional antenna rib structure will be modified for use in Mariner R. The feed is to be changed from a linearly polarized feed to a circularly polarized feed to thereby pick up some needed communication system gain. The directional antenna is to be pointed towards the earth in the same manner as RA-1 and RA-3. The command antennas will be located one on the back of one of the solar panels and one on the front as in the Mariner A.

2. L-band Transponder.

The transmitter-receiver to be used will be the L-band transponder designed for Mariner A. This system is repackaged version of the RA-3 transponder; it is lighter, has a more efficient receiver and more efficiently uses the available transmission power. This unit will transmit three watts over the omniantenna or the directional antenna and receive through the command antennas.

3. Data Encoder

The data encoder will be a stripped down version of the Mariner A data encoder. It will employ the pseudo-noise technique developed for long range life telemetry with a capability of 16 bps at Venus distance. To increase the assurance of success the bit rate at Venus distance will be set at 8 bps.

4. Command Subsystem

A modified Mariner A pseudo-noise type command system is to be used. Eleven ground commands will be provided. Eight are of the real time variety with an actuating pulse being generated. The remaining three are for generation of the midcourse maneuver.

5. Attitude Control

The Mariner A attitude control system will be used, with simplifying modifications. The attitude control subsystem will orient the solar panels towards the sun and point the directional antenna towards the earth. If the attitude is disturbed, the attitude control system will reacquire the sun and earth. The simplifying modifications consist of eliminating the short range earth sensor. This leaves the long range earth sensor to establish pointing of the high gain antenna. If the long range earth sensor is unmodified, earth acquisition will not be possible until about 5-8 days after liftoff. Goldstone should be able to receive the spacecraft omniantenna transmission up to the 5-8 day period.

6. Power

As most of the subsystems to be used are modified Mariner A, the electronics of the power subsystem will be Mariner A providing a 50 volt, 2400 cps square wave as the basic power source. The battery will be a Mariner type battery. Modified MA solar panels will be used. The power profile is shown in Figure II-3 and Figure II-4.

7. Central Computer and Sequencer

A simplified RA-3 CC&S will be used. A modified module will be added to provide cyclic functions and provide switching events up through Venus encounter.

8. Thermal Control

The thermal control of the vehicle will be accomplished by marrying the passive RA-3 thermal control to the semi-active Mariner A shutter technique.

9. Structure

The basic RA-3 hex will remain as is. A new omniantenna support will be added as well as a new support for the long range earth sensor. Other items to be designed are sun sensor mounts, and science supports. Some packaging changes will take place also.

10. Midcourse Propulsion

The midcourse propulsion system for the Mariner R vehicle will be essentially a Ranger RA-3 50-pound thrust monopropellant hydrazine unit. Modifications to the RA-3 design will consist primarily of surface finish changes to tankage, etc., to accommodate the Mariner R thermal control requirements, the substitution of nitrogen for helium as the pressurizing media to better accommodate the six day launch to midcourse firing system storage requirement, and slight plumbing, i.e., envelope changes so that the propulsion system can be loaded into the spacecraft through the bottom rather than the top of the hex as in Ranger. The unit will be capable of imparting a maximum velocity increment of 60 meters/sec to a 460-pound spacecraft.

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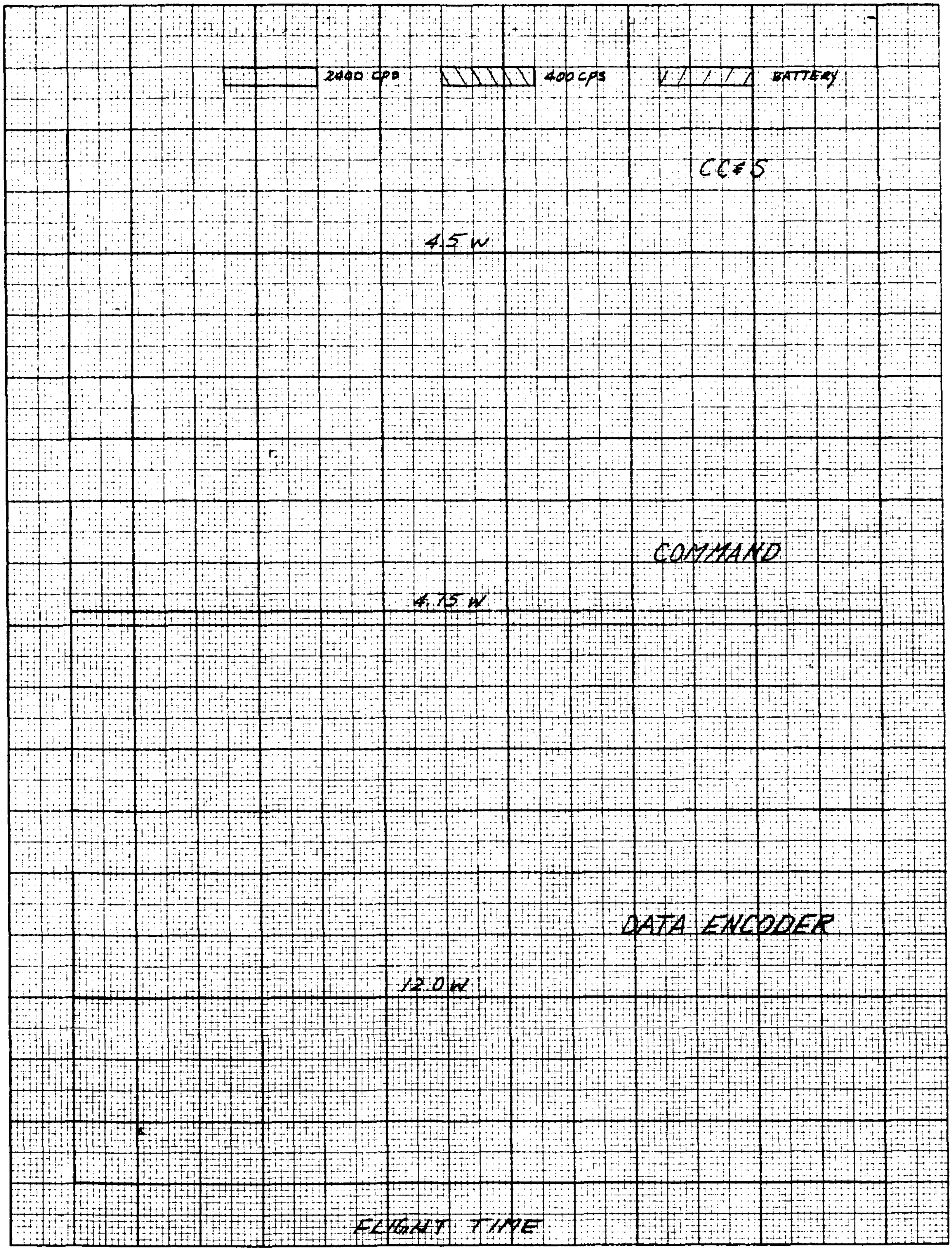

Figure II-3. Mariner R Subsystem Power Profiles

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[redacted]

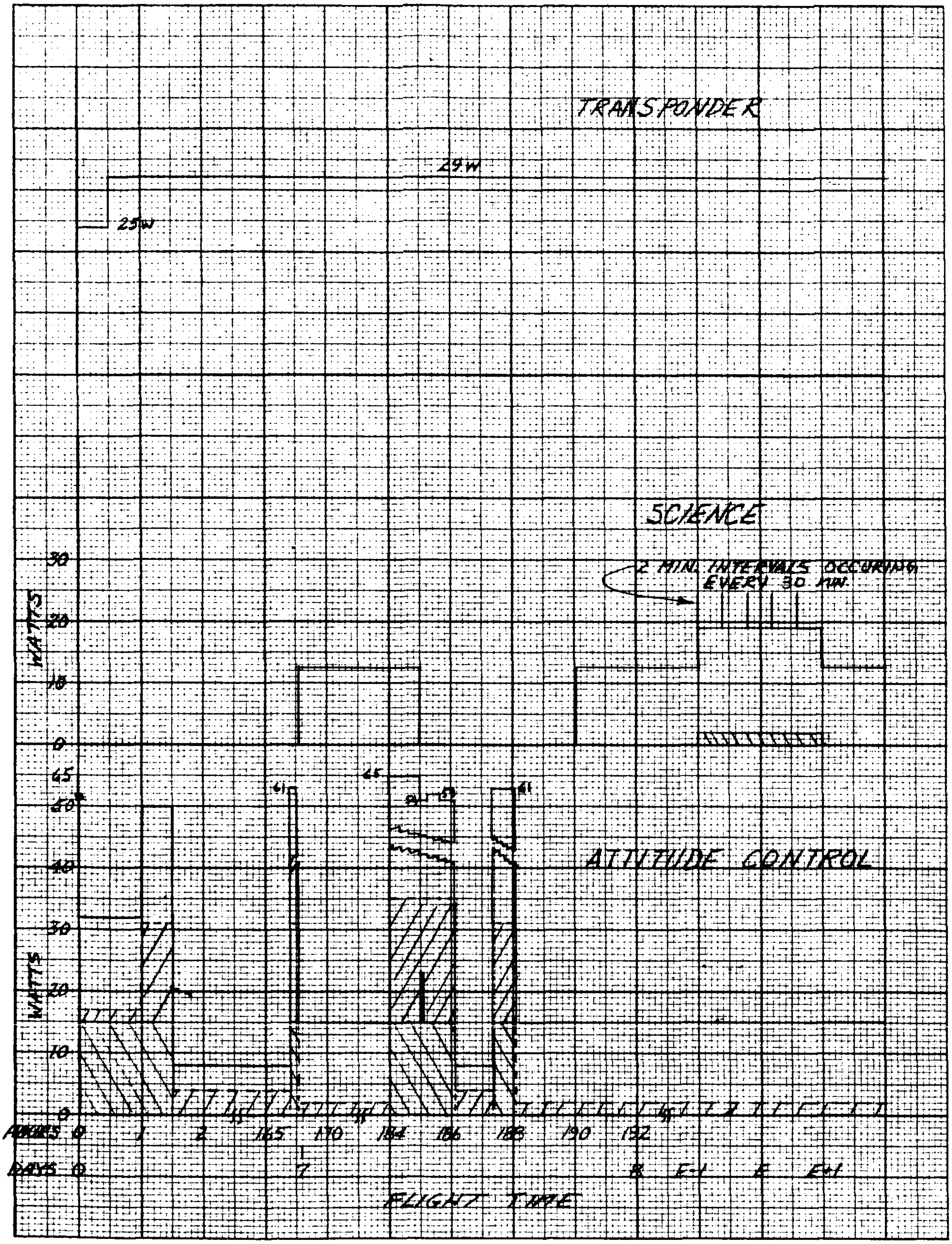

Figure II-4. Mariner R Subsystem Power Profiles

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11. Space Science

The following list is representative of experiments which may be included in the Mariner R Spacecraft. The list is subject to the approval of the National Aeronautics and Space Administration.

a. Planetary oriented experiments

1). A microwave radiometer experiment to measure the brightness temperature of the surface of the planet at 19 mm, and a determination of the presence or absence of water vapor in the atmosphere of the planet at a wavelength of 13.5 mm. A measurement will be made to determine if there is any phase effect which will verfiy or refute some of the atmospheric models which have been postulated.

2). An infra-red radiometer is being considered at a frequency of 8 to 9 microns to determine the granularity of the cloud layer, and if relatively large breaks in the clouds exist, can look to the surface if there is no infra-red absorption by the atmosphere in the window provided.

b. Experiments in the vicinity of the planet

1). A three axis fluxgate magnetometer to measure the three mutually perpendicular components of a magnetic field, if any, in the region about Venus.

2). A Geiger-Mueller counter to determine if there are any regions of trapped particles in the vicinity of Venus.

3). An ionization chamber to measure the integrated charge of any regions of trapped particles in the vicinity of Venus.

4). A cosmic dust experiment to determine if cosmic dust exists in a region close to the planet.

c. Experiments in interplanetary space

1). The magnetrometer mentioned in b. 1). to measure magnetic fields which exist in interplanetary space.

2). A curved plate electrostatic Analyzer (plasma detector) to measure details of the solar corona in interplanetary space in the low energy range.

3). The ionization chamber mentioned in b. 3). to measure the integrated charge of particles in interplanetary space.

[redacted]

4). The cosmic dust experiment of b. 4). to determine the density and distribution of cosmic dust in interplanetary space.

The equipment in items b and c do not represent duplication of equipment, but utilize the same equipment for duplicate functions.

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C. AGENA/MARINER R WEIGHT STATUS

Some 110 pounds appear to be removable from Agena B which will allow an Agena burnout weight of about 1400 pounds, a spacecraft of about 460 pounds, and a spacecraft support structure of about 72 pounds to be injected on an 85 nautical mile orbit to a twice geocentric energy per unit mass value of $.115 \times 10^8$ m^2/sec^2, thus giving a launch period of some 47 days.

The current weight allocations for spacecraft subsystems are shown in Table II-1.

Table II-1. Spacecraft Weight Allocations

Subsystem	Weight Allocation (pounds)
Transponder	41.07
Command	10.00
Power	108.39
Central Computer and Sequencer	9.96
Data Encoder	15.50
Attitude Control	57.40
Structure	82.30
Actuators	3.40
Pyrotechnics	3.75
Motion Sensors	1.33
Spacecraft Wiring	33.00
Propulsion	31.18
Thermal Control	17.00
Space Science	40.00
Contingency	5.72
TOTAL	460.00

[redacted]

SECTION III

OPERATIONAL DESCRIPTION

A. FLIGHT SEQUENCE

1. The flight sequence describes the basic functions of the mission, and no attempt is made to present the detail required for actual design. However, the general outline and command structure is complete.

2. The sequence results from a blending of Ranger and Mariner capabilities with the addition of certain new features to increase mission reliability and capability. The new items, repeated here for clarity are:

   a. A timing sequence contained in the CC&S will start the Venus encounter experiments. A ground command will be available as a back up to initiate the sequence if the CC&S timing sequence does not occur.

   b. As the spacecraft nears Venus a simple scan will allow the planet oriented experiment to acquire the appropriate data.

3. The flight sequence is shown in the following Table III-I

Table III-1. Sequence of Events

Mode	Event	Time	Source	Destination	Comments
Launch to Injection	1. Lift off	T			
	2. Spacecraft injection	T+ 21 to 35 min			
Acquisition	3. Spacecraft separation	Event 2 + 2.6 min			
	a. Transmitter power up		Separation Connector	L-band	
	b. Arm pyrotechnics				
	c. Enable CC&S		Separation Connector	CC&S	
	4. a. Unfold solar panels	T+ 44 min	CC&S	Pyro	Fixed time
	b. Unlatch radiometer				
	5. Turn on attitude control	T+ 60 min	CC&S	Attitude Control	Fixed time
	a. Extends antenna				
	b. Activates sun sensor system				
	c. Activates gas jet system				
	d. Commences automatic sun acquisition				
	6. Sun acquisition complete	T+ 60 to 90 min			
	a. Turn off gyros				
	7. Remove inhibit on automatic earth acquisition	T+167 hrs.	CC&S	Attitude Control	
	a. Starts roll search				
	b. Turns on gyros				
	c. Decrease data rate		CC&S	Encoder	

Table III-1. Continued

Mode	Event	Time	Source	Destination	Comments
LAUNCH	8. Roll override command RTC1		Command	A/C	As necessary throughout flight
	CW hinge override command RTC2				
	CCW hinge override command RTC3				
	9. Earth acquisition complete	Event 7 + 0 to 30 min			
	a. Roll search stops				
	b. Hinge servo starts				
	c. Gyros stop				
	d. Turn on cruise science		A/C	Science	
	e. Switch L-band from omni to directional antenna		A/C	L-band	
	10. Switch L-band to omniantenna RTC4		Command	L-band	As necessary throughout flight
	Switch L-band to directional antenna RTC5				
	11. Complete tracking and send trajectory correction commands	T + 8 days	Command	CC&S	
	a. Roll turn polarity and duration SC1				
	b. Pitch turn polarity and duration SC2				
	c. Velocity increment SC3				
	12. Switch L-band to omni-antenna RTC4		Command	L-band	

Table III-1. Continued

Mode	Event	Time	Source	Destination	Comments
Midcourse Maneuver	13. Initiate midcourse maneuver RTC6	T+ 8.1 days	Command	CC&S	
	14. Start propulsion sequence	Event 13+ 0 min			
	a. Turn on accelerometer		CC&S	A/C	
	b. Turn on gyro		CC&S	A/C	
	c. Turn off cruise science		A/C	Science	
	15. a. Turn off earth sensor power	Event 13 + 60 min	CC&S	A/C	
	Inhibit earth acquisition				
	b. Connect roll gyro capacitor				
	c. Set roll turn polarity				
	d. Exit antenna				
	e. Start roll turn				
	16. Stop roll turn	Event 15+ 0 to 8.7min	CC&S	A/C	
	17. a. Turn on autopilot	Event 13+ 72 min	CC&S	A/C	
	b. Switch out sun sensor pitch and yaw error signals				
	c. Connect pitch and yaw gyro capacitor				
	d. Set pitch turn polarity				
	e. Start pitch turn				
	18. Stop pitch turn	Event 17+ 0 to 16.7 min	CC&S	A/C	

Table III-1. Continued

Mode	Event	Time	Source	Destination	Comments
Cyclic Functions — Midcourse Maneuver	19. a. Start accelerometer integration	Event 13+ 94 min	CC&S	A/C	
	b. Command motor ignition		CC&S	Pyro	
	20. Command motor shutoff	Event 19+ 0 to 2.5min	CC&S	Pyro	
	21. a. Turn off autopilot	Event 13+ 98 min	CC&S	A/C	
	b. Switch out gyro capacitors				
	c. Command antenna to reacquisition position				
	d. Relinquish CC&S control of gyro power and accelerometer				
	e. Commence automatic sun acquisition				
	f. Switch in sun sensor error signals				
	22. Sun acquisition complete	Event 21+ 0 to 30 min			
	a. Turn off gyros				
	b. Turn on cruise science		A/C	Science	
	23. Remove inhibit on earth acquisition	Event 13+200 min	CC&S	A/C	
	a. Turn on earth sensor power				
	b. Turn on gyros				
	c. Initiate roll search				
	d. Turn off cruise science		A/C	Science	

Table III-1. Continued

Mode	Event	Time	Source	Destination	Comments
Midcourse Maneuver	24. Earth acquisition complete	Event 23+ 0 to 30 min			
	a. Switch L-band to directional antenna		A/C	L-band	
	b. Turn off gyros				
	c. Turn on cruise science		A/C	Science	
	CRUISE				
Encounter	25. Begin encounter sequence	Event 27 -1 day			
	a. Turn on encounter science		CC&S	Science	
	b. Switch to encounter T/M mode		CC&S	Data Encoder	
	26. Start encounter operation backup RTC7				
	a. Turn on encounter science		Command	Science	
	b. Switch to encounter T/M mode		Command	Data Encoder	
	27. Nearest approach to Venus				
	28. Return to cruise operation	Event 27 + 1 day			
	a. Switch to cruise science		CC&S	Science	
	b. Switch to cruise T/M mode		CC&S	Data Encoder	
	29. Return to cruise operation backup RTC8				
	a. Switch to cruise science		Command	Science	
	b. Switch to cruise T/M mode Reduce Data Rate		Command	Data Encoder	Backup Function

Table III-1. Continued

Mode	Event	Time	Source	Destination	Comments
Cyclic Functions	Cyclic Function				
	Hinge reference angle update	Every 16.7 hrs	CC&S	A/C	Occurs Throughout Flight
	Science timing pulse	Every 16.7 hrs	CC&S	Science	
	38.4 KC sync		CC&S	Power	

[redacted]

SECTION IV

ANALYSIS - TRAJECTORIES AND GUIDANCE ACCURACY

A. INTRODUCTION

The preliminary analysis of flying a Mariner R spacecraft to Venus in 1962 has been completed. From the results presented, one can deduce how much geocentric energy is required for various cytherean firing periods and observe the trajectory characteristics associated with launch date. Preliminary results concerning the midcourse maneuver are also presented.

B. FIRING PERIOD AND FIRING WINDOW

Figure IV-1 shows a plot of difference in injected weight and C_3, twice the total total geocentric energy per unit mass or the vis viva integral, vs. launch date to Venus in 1962. The minimum energy trajectory to Venus in 1962 has a C_3 of $.087 \times 10^8$ m^2/sec^2. To utilize a firing period greater than one day, this required C_3 must increase, and thus payload capability decrease. Note that for a C_3 of 0.115×10^8 m^2/sec^2, the permissible firing period is from July 24 to September 9, 1962.

On any given launch day, the capability must exist to launch over a period of time. This period of time may be 60 min., 120 min., etc. Unless compensated for, a delay in the launch time will cause an error at Venus encounter. This error increases as the delay becomes longer. The method to be used for compensation of the delay involves simultaneous altering of the launch azimuth and the parking orbit time between Agena burning periods. Southeast launchings are preferred on any given launch day in order to obtain maximum tracking from the Deep Space Instrumentation Facility (DSIF). At present, a launch azimuth span of 96° - 108° east of north is used for the Ranger 3 mission. For this span of launch azimuths, a firing window of 60 to 90 minutes will exist for the proposed Venus launchings. Subject to range safety, this launch azimuth span might be increased to vary from 90° - 114°. If this greater span of launch azimuths is used, a firing window of 150 to 200 minutes will exist. Operational requirements associated with the launching of the spacecraft require a minimum 60 min. firing window. It is very desirable, however, that a 120 minute firing

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window be available on each permissible launch day.

C. FEASIBLE TRAJECTORIES

1. 47-Day Firing Period

For a C_3 of 0.115×10^8 m^2/sec^2, a 47-day maximum firing period will exist from July 24 to September 9, 1962. Arrival dates from December 8 to December 17 must be used to maximize the firing period. The Earth-Venus communication distance on December 8 is approximately 52 million kilometers and for December 17 is 60 million kilometers. Flight times will vary from 139-97 days over the firing period from July 24 and September launchings respectively. As examples of launch times, on August 4 and September 4 the following launch times are required for the associated launch azimuths.

Launch Date = August 4, 1962

Launch Azimuth = 90° East of North, Launch Time = 2:30 AM, EST
Launch Azimuth =108° East of North, Launch Time = 4:30 AM, EST
Launch Azimuth =114° East of North, Launch Time = 5:40 AM, EST

Launch Date = September 3-4, 1962

Launch Azimuth = 90° East of North, Launch Time =11:20 PM, EST
Launch Azimuth =108° East of North, Launch Time = 1:20 AM, EST
Launch Azimuth =114° East of North, Launch Time = 2:20 AM, EST

Geocentric injection will occur over the South Atlantic Ocean and South Africa for the proposed trajectories.

2. Other Firing Periods

From Figure IV-1 and Tables IV-1 and IV-2, one can deduce the required geocentric energy and the corresponding trajectory characteristics for various launch periods. For every 0.01×10^8 m^2/sec^2 change in C_3, payload capability will change by approximately 32 pounds. Geocentric injection will still occur over the South Atlantic Ocean and Southern Africa for the trajectories associated with the larger firing windows.

3. Variation of Earth-Probe-Sun Angle and other Parameters with Launch Date and Time.

Table IV-2 lists the approximate value of the Earth-Probe-Sun angle at various times during flight to Venus for specific launch days. The maximum

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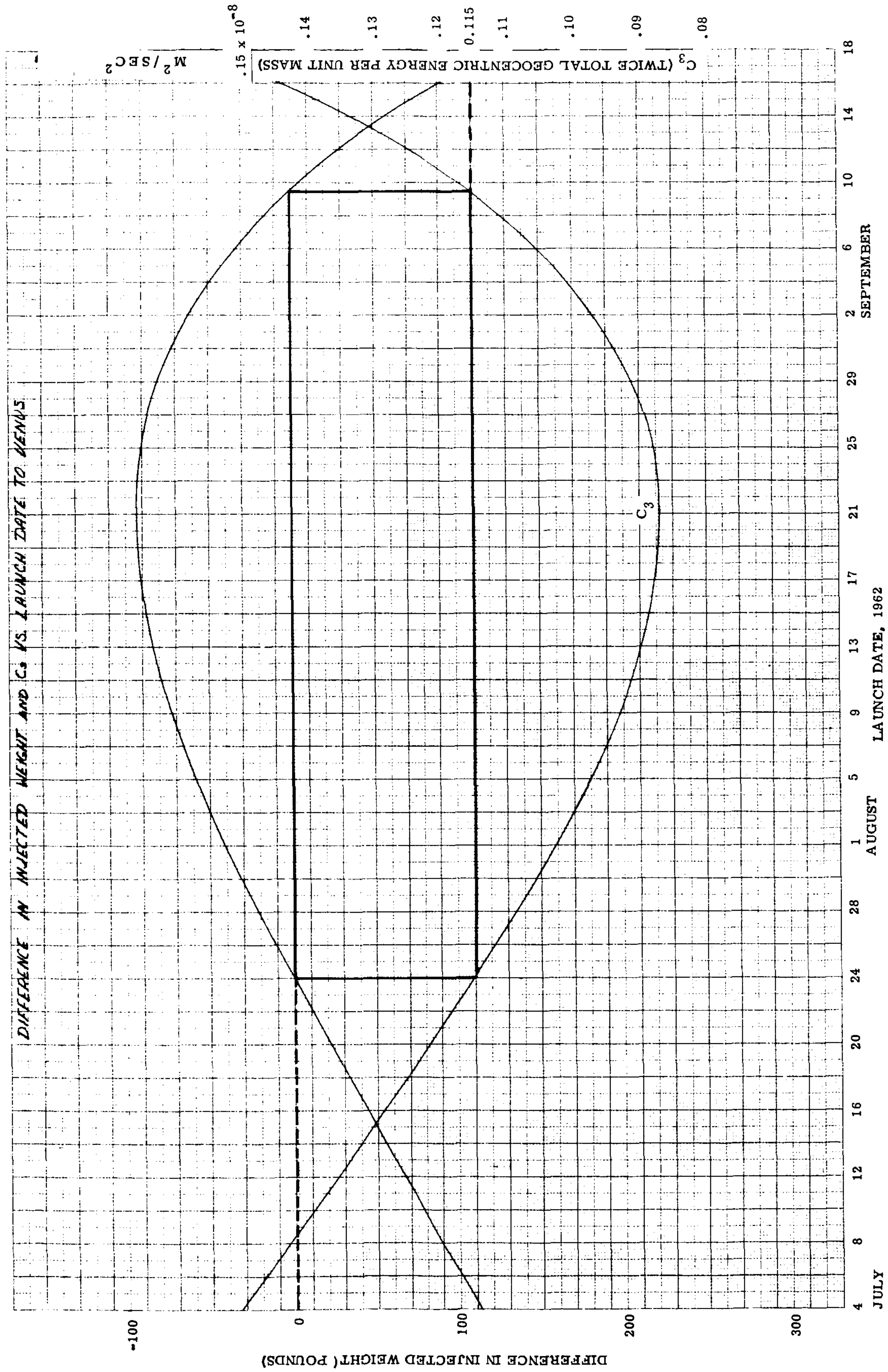

Fig. IV-1. Difference in Injected Weight and C_3 vs. Launch Date to Venus

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Table IV-1. Trajectory Characteristics

$C_3 \times 10^{-8}$ m^2/sec^2	Launch Dates 1962	Firing Period (days)	Flight Times (days)	Earth-Venus Distances at Encounter (millions of km)	Arrival Dates 1962
.087	Aug. 24	1	114	58	Dec. 15
.09	Aug. 13 to Aug. 28	15	124 - 109	58	Dec. 14, 15
.10	Aug. 4 to Sept. 4	31	129 - 101	53 - 58	Dec. 10 - 15
.11	July 27 to Sept 8	43	136 - 98	52 - 60	Dec. 8 - 17
.12	July 21 to Sept. 11	52	142 - 95	52 - 60	Dec. 8 - 17

[redacted]

Table IV-2. Earth-Probe-Sun Angle for Various Launch Days

Launch Day	* Approximate ∢ E-P-S at 3 hours after launch	* Approximate ∢ E-P-S at 6 hours after launch	* Approximate ∢ E-P-S at 2 days after launch	Maximum ∢ E-P-S During Flight to Venus	* Approximate ∢ E-P-S at Venus Encounter
July 21	67°	62°	59°	170°	130°
July 27	71°	66°	63°	170°	130°
Aug. 3	78°	73°	70°	170°	128°
Aug. 13	91°	86°	83°	170°	128°
Aug. 17	96°	91°	88°	170°	125°
Aug. 24	107°	102°	99°	170°	123°
Aug. 28	112°	107°	104°	170°	123°
Sept. 4	125°	120°	117°	170°	123°
Sept. 8	133°	128°	125°	170°	121°
Sept. 11	138°	133°	130°	170°	121°

∢ E-P-S is the Earth-probe-sun Angle, angle between the Earth probe radius vector and the sun-probe vector.

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Earth-Probe-Sun Angle occurs some 30-35 days before Venus encounter. The Earth-Probe distance (from center of Earth) at three hours after launch is equal to 72 thousand kilometers, six hours out equal to 114-120 thousand kilometers, and six days out equal to 1.8 to 2.1 million km.

Most of the important trajectory characteristics are presented in EPD No. 38, Preliminary Standard Trajectory of Mariner A, P-37 and P-38 Missions. The proposed Mariner R trajectories in this document are very similar to those of EPD No. 38. The important differences are that the Mariner R trajectories have arrival dates of December 8 through 17, depending on launch, while the arrival dates for Mariner A trajectories were December 5 and 10th, and that the firing period of the Mariner R mission will be less than the 72 days required for Mariner A.

D. MIDCOURSE GUIDANCE

The dispersion at Venus due to injection guidance errors (assuming no midcourse guidance) has been computed using the following assumptions:

1. Trajectories 2-5 in EPD-38 are representative for Mariner R missions.
2. A covariance matrix of injection errors for the Ranger 3 mission is representative. (This is a valid assumption since the preinjection trajectories are quite similar.)

Figure IV-2 shows a typical 40% dispersion ellipse. The miss components m_1 and m_2 are measured in a plane normal to asymptote of the approach hyperbola. m_1 is in the ecliptic plane. The rms time of flight variation is about .4 days.

Listed below are a number of Agena error sources and their contributions to the total variance of miss.

Error Source	Percent of Variance
Agena thrust misalignment	55
Atlas-Agena mating	16
Agena motor performance	9
All others	20

As can be seen from the above list, there is one major error source which contributes more than half of the total variance of miss.

Because of earth sensor limitations, attitude control about the roll axis cannot be achieved until the 5th or 6th day after injection. Allowing one day

[redacted]

[redacted]

to accomplish attitude stabilization, the midcourse maneuver will be executed on the 7th or 8th day after injection.

In computing the midcourse velocity increment required to correct the dispersion in Figure IV-2, the following additional assumptions are made:

3. Miss (i.e., m_1 and m_2) and flight times are controlled.
4. 10% more propellant is required to make the correction on the 7th or 8th day than on the 2nd day. (Since no differential coefficients were immediately available for the 7th or 8th days, it was necessary to interpolate between available guidance times.)

On each of the trajectories studied an rms maneuver of about 18 m/sec is required. Assuming that the correcting velocity components are spherically distributed a velocity capability of $3.37 \times \frac{18}{\sqrt{3}} = 35$ m/sec is required to correct 99% of all injection errors. If the distribution is needle-shaped, $2.58 \times 18 =$ 46 m/sec is required. A capability of 45 m/sec is considered to be the minimum acceptable for this mission. This capability does not allow for any degradation in the Agena accuracy.

The midcourse motor and associated equipment will be the same as that used on Ranger 3. The Ranger 3 propellant tanks are sized for a maximum capability of 44 m/sec. Thus, if the tanks were full, the correction capability for Mariner R would be

$$\frac{725}{460} \times 44 \cong 69 \text{ m/sec}$$

It is interesting to compute the weight of midcourse fuel required for each m/sec of correcting velocity. Assuming an I_{sp} of 230 sec,

$$\frac{460}{9.8 \times 230} \cong .2 \frac{\text{lbs.}}{\text{m/sec}}$$

The midcourse guidance accuracy for Mariner R should not differ appreciably from the accuracy of Mariner A. In Mariner A, pointing and shutoff errors cause an rms dispersion of 3000 km at Venus. The rms orbit determination error is about 4000 km. Thus the total rms dispersion is about 5000 km. Make the conservative assumption that the miss is circularly distributed with an rms value of 5000 km in any direction. Then 99% of all miss vectors will lie within a circle of radius $3.03 \times 5000 \cong 15{,}000$ km. The circle is drawn to scale in Figure IV-2. The dispersion circle can be centered at any desired point in Figure IV-2. The rms flight time error is about 900 sec. This

[redacted]

corresponds to an rms dispersion of about 5000 km normal to the plane of m_1 and m_2. These guidance accuracies appear to be compatible with the requirements of the scientific instruments.

In order to avoid sterilizing the Agena vehicle it is necessary to ensure that the probability of the Agena hitting the planet is less than .1%. Two methods are being considered. In the first method (used in Mariner A) the Agena is aimed at the planet, but shortly after injection a small retro rocket fires on the Agena and causes it to miss. An impulse of 2 m/sec is required in (or opposite to) the direction of motion.

The alternative method is to aim the Agena so that it will miss the planet and correct the trajectory of the spacecraft with the midcourse maneuver. A midcourse impulse of 15 m/sec is required to correct for the bias.

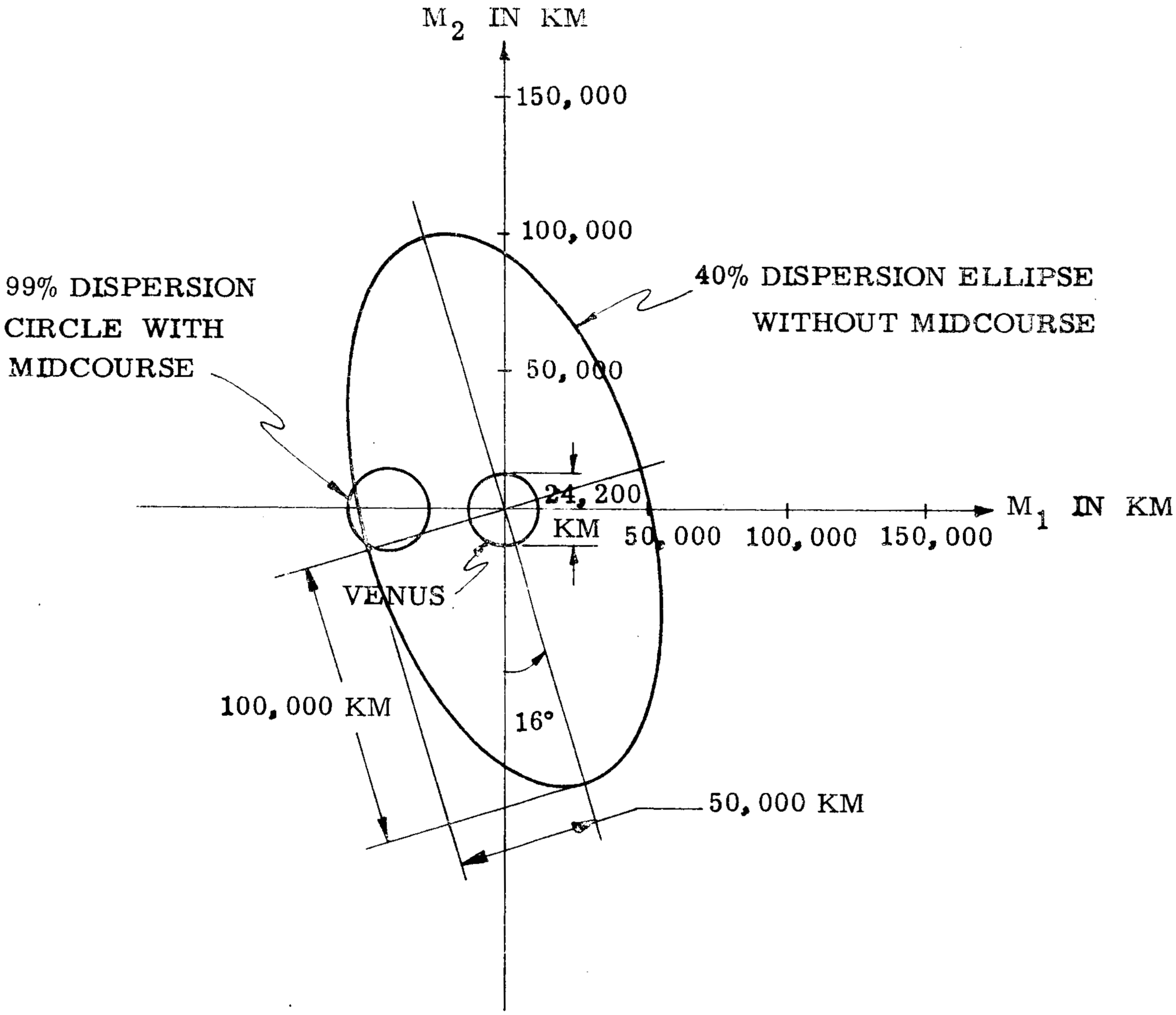

Fig. IV-2. Dispersions at Venus